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OVERCOMING GENESIS MISSION DESIGN CHALLENGES

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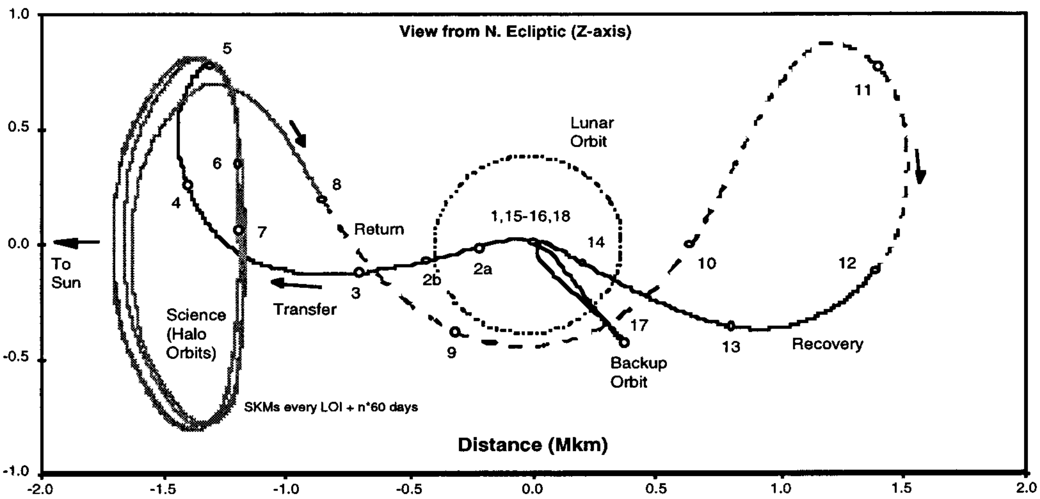
ABSTRACT

Genesis is the fifth mission selected as part of NASA's Discovery Program. The objective of Genesis is to collect solar wind samples for a period of approximately two years while in a halo orbit about the Sun-Earth colinear libration point, L1, located between the Sun and Earth. At the end of this period, the spacecraft follows a free-return trajectory with the samples delivered to a specific recovery point on the Earth for subsequent analysis. This type of sample return has never been attempted before and presents a formidable challenge, particularly with regard to planning and execution of propulsive maneuvers. Moreover, since the original inception, additional challenges have arisen as a result of emerging spacecraft design concerns and operational constraints. This paper will describe how these challenges have been met to date in the context of the better-faster-cheaper paradigm.

MISSION OVERVIEW

The trajectory for the Genesis mission[1], shown in Figure 1, was the first to be designed using modern dynamical systems theory[2]. The mission is scheduled for launch in January-February 2000, using a Boeing Delta II launch vehicle with a Star 37 third stage. The spacecraft will experience a low energy injection (maximum C3 of -0.6 km^2/s^2) into a specially determined orbit transfer to L1 which requires about three or four months, depending on launch date. A unique feature of the mission is that it requires only one deterministic maneuver, which inserts the spacecraft into the Lissajous or halo orbit after the transfer. A number of trajectory correction maneuvers (TCMs) and about 13 halo orbit station keeping maneuvers (SKMs) are anticipated as well, as indicated in Figure 1. Genesis will spend a minimum of 22 months collecting samples of the solar wind and taking science data, mostly during the halo orbit. After four loops near L1, the spacecraft will return freely past Earth, sweep around the L2 libration point (on the far side of the Earth from the Sun) and be guided to a specific target that results in a daylight parachute recovery at the Utah Test and Training Range (UTTR) near Salt Lake City beginning in August 2003. A single trajectory from Lissajous orbit insertion (LOI) through return and recovery may be used for prime mission launch opportunities in January 2000, with an additional trajectory available to accommodate February launch opportunities. Diversion into a 19-day backup orbit can be performed in the event that conditions are not favorable for recovery on the first attempt.



Key Events (by Earliest Date/Mission Day)

1. Launch	07jan01/000	5. Start Science	20mar01/072	10. TCM-7	24apr03/837	15. Entry	16aug03/951
2a. TCM-1a	08jan01/001	6. TCM-4	20apr01/103	11. TCM-8	10jun03/884	or	
2b. TCM-1b	10jan01/003	7. LOI (TCM-5)	30apr01/113	12. TCM-9	17jul03/921	16. Divert	16aug03/951
3. TCM-2	14jan01/007	8. End Science	28mar03/810	13. TCM-10	06aug03/941	17. Apogee	26aug03/961
4. TCM-3	11feb01/035	9. TCM-6	10apr03/823	14. TCM-11	15aug03/950	18. B. Entry	04sep03/971

Figure 1. Genesis Mission Trajectory for Prime Mission (January 2000 Launch)

OVERVIEW OF SPACECRAFT DESIGN AND CONSTRAINTS

To achieve a level of cost-effectiveness consistent with a Discovery-class mission, the Genesis spacecraft design was adapted to the maximum extent possible from designs used on earlier missions, such as Stardust, another sample return mission. The spacecraft consists of a bus, including two solar arrays deployed after separation from the launch vehicle, and Sample Return Capsule (SRC) with the science payload, as shown from two perspectives in Figure 2.

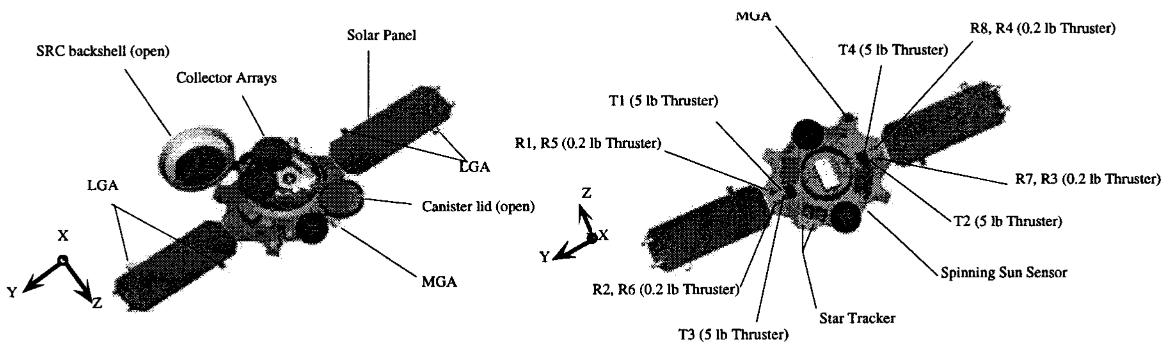


Figure 2. Forward Deck View (Normally Pointing Toward Sun) and Rear Deck View

Total spacecraft mass at injection is 646 kg, which can accommodate a maximum fuel load of about 144 kg (total mission Δv exceeding 530 m/s). Thrusters (four 5 lbf directed in the +X direction) and eight 0.2 lbf canted at 45° away from +X towards +Z or –Z)) form part of a hydrazine fueled blowdown system and are located on the opposite side of the spacecraft from the SRC to minimize contamination of samples. Over the course of solar wind collection, when the SRC backshell is open and various science collection instruments deployed, fuel expenditure is limited to a total of 30 kg. The SRC backshell must be closed with all science instruments in a stowed configuration whenever the 5 lbf thrusters are employed. Furthermore, during developmental testing, it has been determined that while one of the science instruments, the concentrator, is deployed, the spacecraft can point no more than 60° off sun to avoid inducing thermal gradients which may irreparably damage this instrument. Upon final return to Earth, the SRC is designed to be released from the spacecraft bus, directly enter the Earth's atmosphere, and descend over UTTR for mid-air retrieval by helicopter. After release of the SRC, the bus can be safely deboosted using remaining fuel to enable descent over the Pacific Ocean and away from populated areas.

The electrical power subsystem (EPS) also includes a 16 amp-hour battery. Except when maneuvers are required, the solar arrays will be pointed generally to within 10° of the Sun to ensure sufficient power, or in the prevailing solar wind direction for science collection and checkout, the latter being 4.5±0.5° ahead of the sun near the Ecliptic Plane. A time limit of about 85 minutes is imposed when the spacecraft can be more than 30° off sun (equivalent to Δv limit of about 110 m/s). The telecommunications subsystem employs S-band uplink and downlink and includes low gain antennas (LGAs) directed both forward and aft plus a medium gain antenna (MGA) pointed in the aft direction.

Spin stabilization was chosen as a simple means of attitude control, in lieu of three-axis stabilization. The spacecraft normally spins at 1.6 revolutions per minute (rpm), but spin rate must be increased to as much as 10 rpm typically when using the larger (5 lbf) thrusters, usually when the required translational Δv exceeds 2.5 m/s. No reaction wheels, gyros or accelerometers are included in the attitude control subsystem (ACS). All attitude changes, including spin changes and precessions, must be performed open loop with thrusters. Because the thrusters do not produce balanced torques, all attitude control maneuvers contribute to the translational Δv and affect the spacecraft trajectory. Consequently, all burn, turn and spin change components must all be accounted for when planning a maneuver. Thruster activity, asymmetric mass properties and misalignments also induce wobble and nutation, and can result in maneuver execution errors as large as 6% or so in many cases.

A star scanner and sun scanners support attitude determination and control. However, only the spinning sun scanner (SSS) can provide reliable attitude data when spinning at greater than 2 rpm. Originally, the star scanner was expected to function as a star tracker and solely provide three-axis spacecraft attitude, at least at lower spin rates (1.6-2 rpm). However, poor performance of the star scanner demonstrated thus far has resulted in the need to develop a hybrid attitude control mode known as spin track, in which both the star scanner and a digital sun sensor (DSS) are needed to ensure three-axis attitude determination. Unfortunately, the field of view of the DSS is limited to about 30° from the Sun, so all maneuvers more than 30° off Sun may have to rely on the SSS alone by default. Moreover, attitudes which place the spin axis near or directly away from the Sun can render the SSS blind, resulting in the need to observe keep-out zones (KOZs) about the sunward or anti-sunward directions to ensure sufficient attitude knowledge. Such KOZs can grow as large as 30° in the presence of nutation. Because the SSS cannot provide complete three-axis attitude information, turns performed while utilizing the SSS are generally limited to precessions along a line of longitude emanating from the Sun.

Performance issues regarding the star scanner performance and concentrator thermal design issues constitute additional challenges not foreseen in earlier phases of the mission design. As a result, three specific phases of the mission have required some level of redesign, as described in subsequent sections. Additional details on spacecraft design are available in referenced documents[1,2,3].

TRANSFER AND LISSAJOUS ORBIT INSERTION

A three-maneuver strategy is envisioned for correction of launch injection errors over the course of transfer out to the LOI point. The first trajectory correction maneuver after injection would likely need to be performed within 24 hours to avoid exorbitant Δv cost. The first maneuver is mainly an energy correction designed to compensate for a post-launch injection underburn or overburn. Successful execution of TCM-1 entails minimizing the number and complexity of post-launch activities, which could potentially trigger fault protection and result in lengthy recovery and delay of this critical maneuver. For this reason, only the SSS will be available for attitude determination within the first 72 hours after injection. This affords adequate time to bring the star scanner on line and to perform a series of scanner calibrations over a period of several days to ensure adequate performance for subsequent maneuvers. However, as stated previously, sunward and anti-sunward KOZs must be observed for the SSS. Unfortunately, the most likely directions for TCM-1 lie either along or opposite to the spacecraft velocity vector, which lies close to the sunward direction not only at 24 hours but over several days beyond launch. To accommodate all constraints, a TCM-1 implementation strategy has been developed, as illustrated in Figure 3.

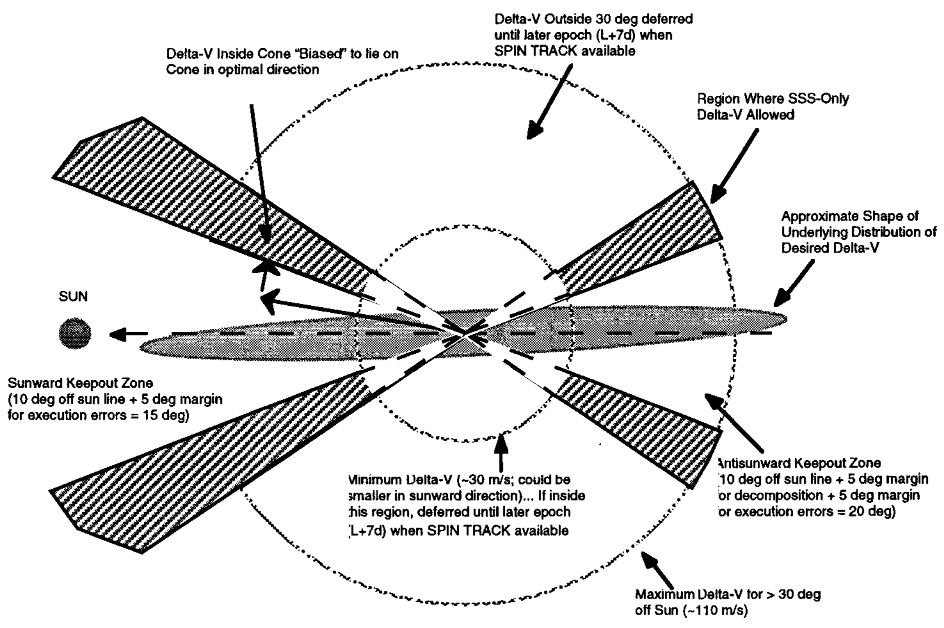


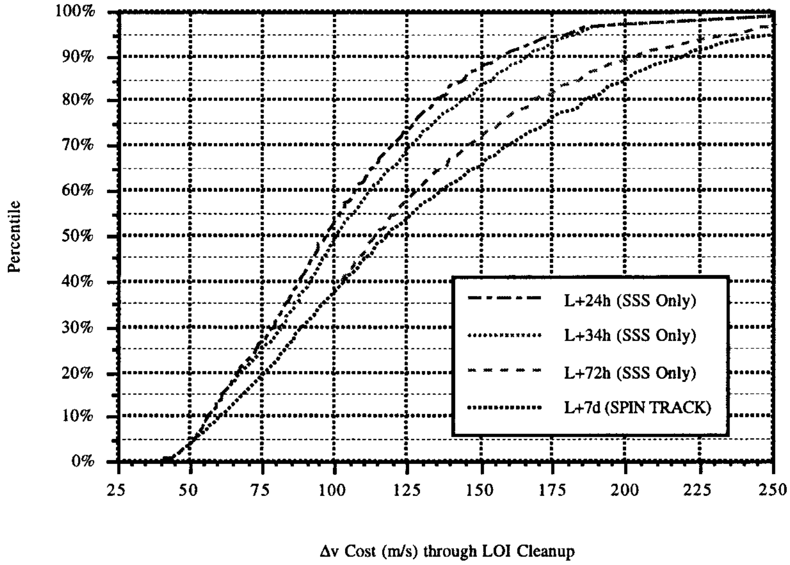
Figure 3. Early TCM Implementation Strategy

This strategy limits the direction of TCM-1 to a specific range of cone angles in either the sunward or anti-sunward direction. If the desired Δv direction lies close to the Sun line between the allowed regions, a cross-track bias is applied to effect a shift to the most optimal Δv direction allowed for the burn in the context of subsequent maneuvers. If the Δv direction lies far off the Sun, the Δv magnitude tends to be small. Whenever Δv magnitude is small enough, TCM-1 at 24 hours after launch (L+24h) can be avoided altogether with the first TCM performed at TCM-2, seven days after launch (L+7d). A minimum of 30 m/s should guarantee that the anti-sunward KOZ is satisfied after decomposition of the maneuver into turn, burn and spin change components is considered. A smaller minimum threshold (e.g., 10 m/s) could be allowed in the sunward direction. It is assumed that the star scanner will be operational for TCM-2, at least at a minimal capability required to support the spin track mode of attitude control described previously.

If Δv magnitude required for TCM-1 becomes large enough (e.g., 110 m/s), particularly in the anti-sunward direction where off-Sun time is limited by power constraints, an earlier maneuver at 15 hours after launch must be considered. To facilitate sequence preparation and execution, such an emergency TCM would need to be pre-tested prior to launch. This requires that the number of possible TCM-1 sequences be reduced to a manageable number of cases. This can be accomplished by selecting one sunward and one anti-sunward direction which are optimal or near-optimal for the vast majority of possible launch injection errors, but allowing for one uploadable parameter, the burn duration, to be determined and uplinked just prior to sequence execution.

To afford more margin in the TCM-1 timeline, it is possible that execution of TCM-1 might be planned for as much as 34 hours after launch. Moreover, the strategy must also allow for the possibility of anomalies with the spacecraft or interruptions in the Deep Space Network (DSN) radiometric tracking coverage which could delay orbit determination (OD) for maneuver planning, as well as uplink and execution of the TCM-1 sequence. It is also possible that a strategy similar to the emergency TCM could be used at 24 hours or later, as a contingency in the event of these or other operational difficulties.

Figure 4 compares the performance in terms of Δv costs associated with various operational scenarios, as obtained from monte-carlo simulation runs which include both OD and maneuver errors, as well as errors arising from launch injection. About 175 m/s were originally budgeted for Transfer and LOI[3]. Nevertheless, the performance as shown remains fairly robust, even if TCM-1 is delayed to as much as 72 hours after launch (L+72h). This is largely due to the addition of 23 kg of fuel, which has extended the mission Δv capability by 80 m/s or more from 450 m/s to the current 530 m/s, including a margin of about 67 m/s. This means that about 250 m/s would be available to support trajectory corrections associated with Transfer and LOI.



NOTE: SSS-only cases include effects of early TCM strategy, particularly delay to L+7d under conditions outlined in Figure 3.

Figure 4. Comparison of Performance for Early TCM Options Examined

LISSAJOUS ORBIT STATION KEEPING

Another challenge for the mission design involves station keeping maneuvers required to keep the spacecraft on the halo orbit during solar wind collection and positioned for eventual return to Earth. During this period, the primary science instrument, known as the concentrator, collects nitrogen and oxygen ions onto a target via electrostatic grids. If the concentrator is pointed more than 60° or so away from the sun when exposed to empty space, the grid becomes shaded introducing a large thermal gradient with respect to the container. This can cause irreparable damage to an instrument whose data return is deemed of the highest priority for the Genesis Mission. If turns away from the sun are unavoidable, alternatives are to close the concentrator lid and SRC backshell or to shade the concentrator with other deployable collection arrays, both of which create an undesirable interruption in solar wind collection with degradation of the quality of samples to be returned to Earth later.

To avoid excessive turns away from the Sun to the maximum extent possible, all station keeping maneuvers are to be biased towards the Sun. A deterministic bias level of 1.5 m/s has been selected, which, with execution errors, adds about another 30 m/s to the Δv budget. This is easily covered by the available margin of 67 m/s and is small enough to be absorbed via reoptimization of the Genesis trajectory without adversely affecting mission requirements, particularly at Earth entry. Conveniently, 1.5 m/s also falls half way between the typical two-way turn circle diameter (0.5 m/s) and the normal maximum maneuver size on the 0.2 lbf thrusters (2.5 m/s)[3,4]. Additional information on the halo orbit station keeping strategy will be provided in another paper to be presented in June 2000[5].

RETURN AND RECOVERY

Finally, to attain the accuracy required for recovery with a high degree of robustness, some of the final maneuvers, in particular TCM-10 at entry minus 10 days (E-10d) and TCM-11 at one day prior to entry (E-1d), must be performed using a highly accurate series of spin rate changes (spin-up followed by spin-down). This approach utilizes the following relationship between Δv, directed along the +X axis, and Δω or change in spin rate for the spacecraft:

$$\Delta v_{spin} \cong \frac{I_{xx}\Delta\omega_x}{rm} \tag{1}$$

The thruster lever arm r is well defined beginning before launch. However, mass properties change over the course of the mission. Because there are no accelerometers on Genesis, Δv cannot be determined by the on-board ACS software. However, through observation of a spin change event from the ground coupled with spin rate estimates available from telemetry throughout the event, the spacecraft mass properties I_{xx}/m can be established very accurately. Studies have demonstrated that reduction of all fixed errors to 3 mm/s and proportional errors to 1% would permit all of the aforementioned entry requirements to be met with reasonable performance margin[3]. Such in-flight calibration of mass properties must be performed prior to the final maneuvers. The best locations for these events are near Sun-Earth line crossings on the Return Phase, beginning around mission days 840 and 910.

Such maneuvers require an extensive period of time to execute, partly because nutation introduced during spin-down requires minutes, or even hours, to damp out. Therefore, it is critical that these maneuvers be directed within 30° of the Sun to avoid violation of power constraints mentioned previously. However, because of KOZs associated with the SSS, it is also important that the Δv direction be kept at least 10-15° away from the Sun, as well. This requires yet another biasing strategy, where an anti-sunward bias is introduced at TCM-9 (E-30d), such that deterministic biases of less than 1 m/s at ~22° off Sun are introduced. This strategy is illustrated in Figure 5.

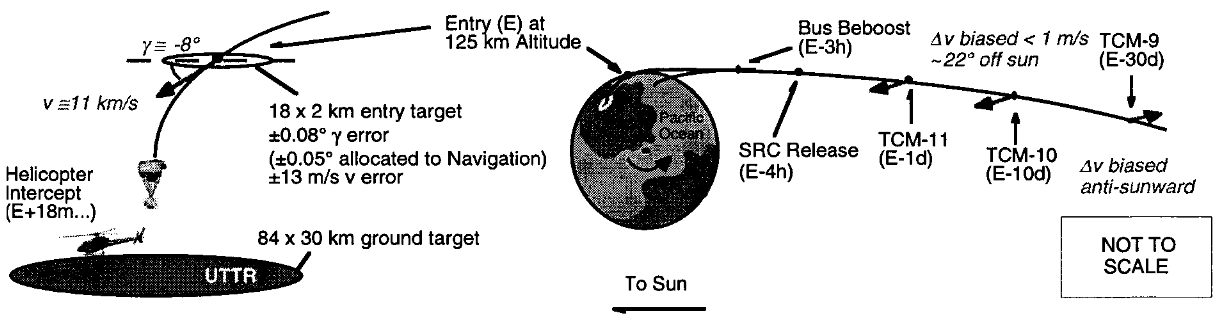


Figure 5. Entry Timeline Indicating Biasing Strategy and Targeting Requirements

Figure 6 indicates the performance determined to date using on the strategy outlined above, as determined by monte-carlo simulation results. Minimum deterministic biasing levels of 0.2 m/s and 0.6 m/s are suggested for TCM-10 and 11, respectively. With these biases and 1% execution errors assumed, entry requirements at the 125 km interface altitude are shown to be easily met.

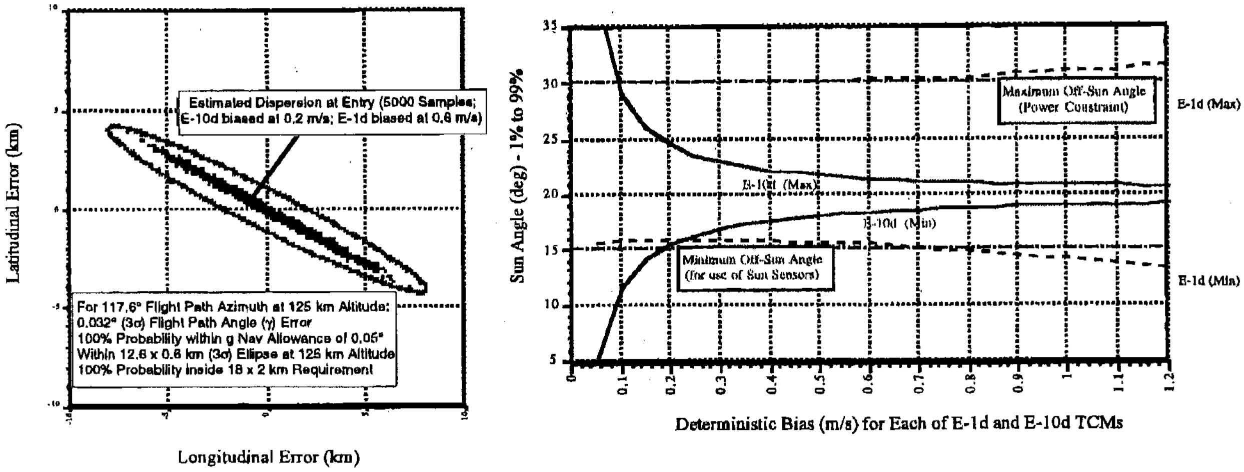


Figure 6. Performance Associated with Biased Entry Maneuvers

CONCLUSION AND FUTURE WORK

The Genesis Mission Design and Navigation has demonstrated considerable resourcefulness and creativity in overcoming challenges arising thus far from the spacecraft design and other mission constraints. The aforementioned strategies and analyses are subject to further refinements as additional information, such as updates to predicted launch injection errors, becomes available. Also, contingency scenarios, including the backup orbit, will be examined in more detail to ensure mission readiness.

ACKNOWLEDGMENTS

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